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14. ABSTRACT Analyses and calculations were carried out to quantify the gain in performance (as measured in terms of thrust specific fuel consumption and range productivity) in aircraft turbine engines incorporating adaptability/variability. A 5-10% variability in (turbine and propelling) nozzle areas would allow the compression system to have an operating point for subsonic loiter identical to that for supersonic flight to destination, resulting in a 20% improvement in thrust specific fuel consumption. Zero spillage engines (engines in which the inlet and the engine capture the full streamtube, at constant area over flight Mach number ranging from 0.8 to 2.5) are shown to be feasible. Progress toward low fuel consumption (i.e. fuel efficient engine technology) can be achieved through engineering very high pressure ratio (~hundreds) compression system with high polytropic efficiency compressor components. Enablers that include the variable area swirling turbine, flow aspiration, compressor rim cooling and intercooling are suggested for realizing the required engine variability, zero spillage engines and fuel-efficient propulsion systems. While these technology enablers are technically challenging in practice, they are sure to pay off handsomely (such as significantly broadening the scope and flexibility of missions presently not accessible with engines of fixed geometry)!

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ADVANCED AND ADAPTABLE MILITARY PROPULSION

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ADVANCED AND ADAPTABLE MILITARY PROPULSION

MIT GAS TURBINE LABORATORY

December 2007

INTRODUCTION

- Adaptable engine technology
  - Engine adaptability (through varying bypass ratio, cycles and geometry) enables new missions not accessible by traditional fixed geometry engine
- Overall goal
  - Assess requirements and feasibility for adaptable engines
  - Determine potentially realizable gains with adaptable engines

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Improvements in aircraft turbine engines may come from traditional approach of improving component efficiency, size, and weight on incremental basis. New cycles and engine arrangements offer paths that have seen much less investment. New engine arrangements can be considered to fall into three categories. The first is increased complexity of fixed cycles (e.g. intercooled arrangement) that could yields improved thermodynamic efficiency or specific power of the engine. The second is variable cycle engines which, strictly speaking, vary the thermodynamic cycle of the engine and include such concepts as such as variable pressure ratio or variable turbine area. The third category includes variable bypass ratio configurations which are used to adjust the propulsive efficiency and specific thrust of the engine, usually as a function of flight speed. Of course, particular engine concepts can include elements of more than one of these categories.

Current engines of fixed geometry are optimized for a single flight Mach number. To appreciate the value that an adaptable engine might bring, one first must understand how a fixed geometry behaves over a range of flight speeds. For instance in a low bypass ratio (0.8) turbofan engine operating at flight Mach numbers ranging from 0.85 to 2.5, the specific fuel consumption, SFC, increases monotonically with flight speed, but the range productivity (flight Mach no. divided by SFC, a measure of how far the airplane can travel per unit of fuel) is a maximum at a about Mach 1.8.

SPECIFIC TECHNICAL TOPICS ADDRESSED IN PRESENT EFFORT

- **Optimum bypass ratio (BPR) cycles for maximum range productivity at minimum specific fuel consumption (SFC)**
- **Identical compression system operating point (OP) for supersonic flight at M=2.0 and effective subsonic loitering at M=0.4 to 0.8**
- **Zero spillage engine (ZSE)**
- **Variable Area Swirling Turbine (VAST)**
- **Fuel-efficient engines and propulsion system for sensor-aircraft**

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Several technical areas of relevance to the technology of engineering an adaptable engines are explored and assessed. The first topic that was examined is quantifying the potential gain achievable in the range productivity for a variable bypass cycle engine in which the bypass ratio at a given flight Mach number is optimized for minimum SFC. The second topic is assessing the potential gain of an adaptable engine and the required variability in turbine nozzle area, propelling nozzle area and core nozzle area to have the compression system operating point and inlet corrected flow for subsonic loiter identical to those for supersonic flight to destination. The third topic is on assessing the concept, the feasibility and the required variability for a zero-spillage engine (ZSE). As will be seen in the results presented here, a required variability for ZSE is a factor of 4 variation in turbine nozzle area. This leads to the exploration and assessment of the concept of variable area swirling turbine (VAST) as an enabler for ZSE. Finally the requirements for the engineering of fuel-efficient engines for sensor-aircraft are addressed on a somewhat quantitative basis.

RANGE POTENTIAL (1)

- Breguet Range equation:

$$\text{Range, R} = \eta_0 \, H \left(\frac{L}{D}\right) ln\left(\frac{W_0}{W_0 - W_{fuel}}\right)$$

  - Engine performance by η_o, the overall efficiency
  - Fuel by H, rate of energy released due to combustion
  - Aircraft performance characterized by lift-to-drag ratio L/D
- Overall efficiency = (thermal efficiency)(propulsive efficiency)
  - Thermal efficiency limited by T_{t3}
  - Propulsive efficiency controlled by bypass ratio BPR
- W_0 the initial gross weight
- W_{fuel} the weight of fuel consumed

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The well-known Breguet Range equation is selected as a metric for quantifying the overall performance of aircraft-propulsion system. The characterization of each of the terms in the range equation is delineated on this vue.

RANGE POTENTIAL (2)

- **Breguet Range equation rewritten as:**

$$R = \left(\frac{3600\,U_0}{sfc}\right)\left(\frac{L}{D}\right) ln\left(\frac{W_0}{W_0 - W_{fuel}}\right)$$

$$= \left(\frac{M_0}{sfc}\right) 3600\,a_0\left(\frac{L}{D}\right) ln\left(\frac{W_0}{W_0 - W_{fuel}}\right)$$

- **Range productivity=** $\left(\frac{M_0}{sfc}\right)$

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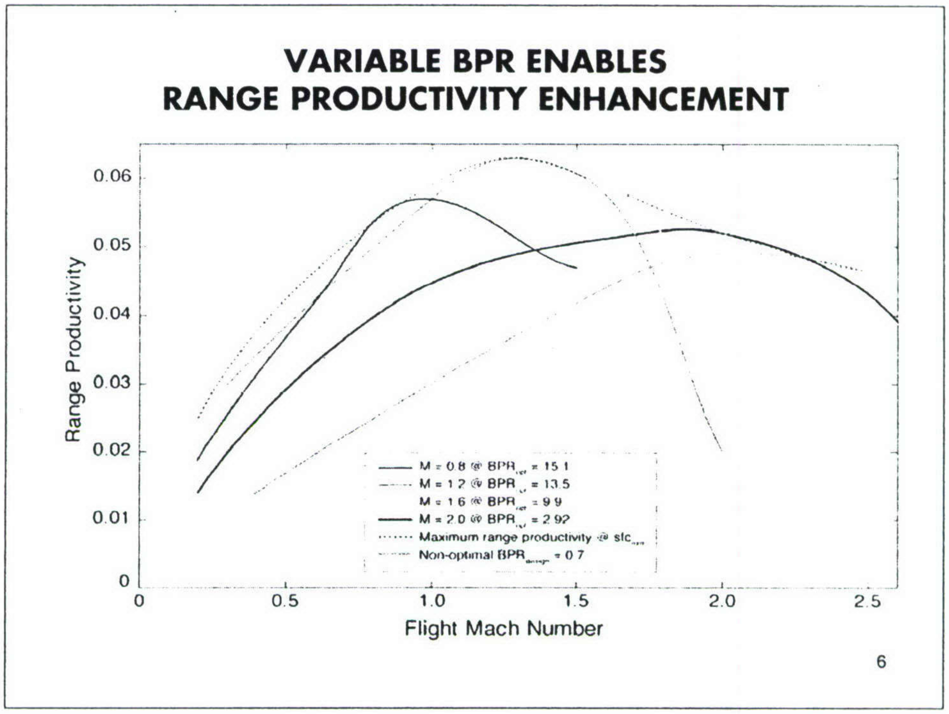
On this vue the Breguet Range equation is cast in terms of the flight Mach number M0, the specific fuel consumption sfc, the ambient speed of sound a0, the lift-to-drag ratio L/D, and the ratio of fuel weight consumed to the initial weight.

The range productivity, given by the ratio of flight Mach number M0 to the specific fuel consumption sfc, provides a measure of how far the airplane can fly per unit of fuel.



An engine with variable fan bypass ratio can provide a factor of 2 to 3 improvement in range productivity (ratio of flight Mach number to specific fuel consumption, a measure of how far the airplane can fly per unit of fuel) compared to a non-optimal engine with a design bypass ratio of 0.7. The dashed curve delineates the envelope of range productivity of cycles with maximum power/unit air flow and with BPR at the minimum SFC.

A major point is that we can currently design engines that achieve near optimum performance at one operating point but that this performance falls-off at other operating conditions. This is especially true when the design point is supersonic.

FAIRLY HIGH BYPASS ENGINE FOR SUPERSONIC FLIGHT

- For cycles with maximum power/unit air flow and with BPR at the minimum SFC
  - M = 2 engine with BPR ~ 3.0 provides good range productivity at design compared to that with BPR ~ 0.7
    - ➢ Margin improvement for cycles with maximum power/unit air flow
    - ➢ Range productivity ~ flat to M ~ 1 and lower with maximum at M ~ 2 for cycles at minimum SFC

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A key implication of computed results displayed on vue 6 is that a fairly high bypass ratio (BPR~3) engine for supersonic flight not only yields enhanced range productivity at design (M=2) compared to that with the usual practice of a low bypass ratio (BPR~0.7) engine but also a range productivity that is flat to M~1 and lower.

SUPERSONIC FLIGHT AT M = 2 TO DESTINATION WITH EFFECTIVE M = 0.8 LOITER

- Achievable with variable
  - Turbine nozzle area
  - Propelling nozzle area
  - Core nozzle area
- For identical compressor OP and inlet corrected flow at M = 2 and M = 0.8
  - Extent of area variation needed ~5-10%
- Significant margin improvement (~20% in TSFC and range productivity)
  - TSFC of 24 gm/kN/s to 29 gm/kN/s for fixed geometry
  - Range productivity of 0.033 to 0.027 for fixed geometry
- New inventions as enablers
  - For example, innovations to engineer variable turbine nozzle

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Cycle analysis have been implemented to quantify the required variability in turbine nozzle area, propelling nozzle area and core nozzle area to have the compression system operating point and inlet corrected flow for subsonic loiter identical to those for supersonic flight to destination. The capability to achieve this would result in significant margin improvement. For instance to achieve a ~20% improvement in TSFC and range productivity would only require an area variation of 5 to 10% in (turbine and propelling) nozzle areas. Clearly the technical challenge lie in devising an innovative way of changing the turbine nozzle area by 10%.or greater

ZERO SPILLAGE ENGINE

- Concept - while delivering the needed thrust, engine
  - Adjusts to always accept the mass flowing through a fixed capture area A_c, independent of flight Mach No., M_0
  - Also use adjustability to minimize mission fuel consumption
- Potential advantages
  - Eliminate spillage drag across flight regime (10-20%)
  - Maximize thermal efficiency at all flight conditions
  - Enables fan pressure ratio variation to increase propulsive efficiency

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A zero spillage engine is one in which the inlet and the engine capture the full streamtube, at constant area, entrained by the inlet lip at maximum M_0, over the range of $0.8<M_0<2.5$. The intent here is twofold: first to capture the influence of the propulsion system on the airplane drag off-design, the second is to do so in a way which is largely independent of the aircraft design. Thus by eliminating spillage drag entirely in this extreme example, we do not need to calculate it's airframe dependant effects.

ANALYTICAL MODEL FORMULATION

- Simplified cycle model
  - Fixed component efficiencies (90% polytropic)
  - Limited changes to gas properties
- Assumptions in these calculations
  - Compressor exit temp (T_{T3}) fixed at 950K
  - Bypass ratio fixed (BPR) fixed at one flight condition
  - Capture area fixed (A_C)
  - Fan inlet area fixed (A_2)
  - Fan pressure ratio fixed set at 3.4 at SLS
  - Turbine inlet temp (T_{T4}) fixed
- Compare model output with thrust required by AFRL aircraft model
  - Adjust model and inputs as needed
- Check points of analytical solution with more detailed cycle deck (GASTURB)
  - Analytical model and cycle deck give similar results.

AIRFRAME REQUIREMENT
-From AFRL-

Flight Mach, M_0	0.8	1.5	2.0	2.5
Altitude (m)	9,150	12,280	15,250	16,775
Thrust (N)	25,568	33,718	34,472	37,418
Aircraft L/D	12	9.1	8.9	8.2
Fixed capture area mass flow (kg/s)	51	68	65	69
Thrust/(airflow * a_0)	1.68	1.66	1.78	1.82

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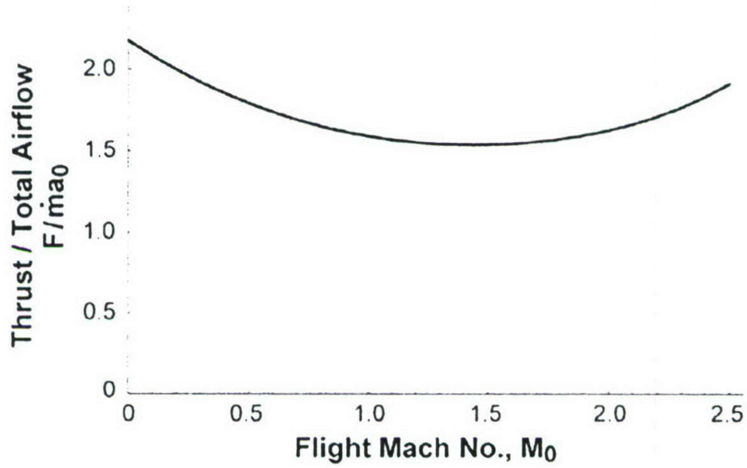
Shown in this table is the airframe requirement (put forward by AFRL) and the derived specific thrust requirements.

The airframe thrust requirement defines the thrust requirement vs (M_0,h)-space, and hence thrust/airflow

The required thrust, together with the airflow set by capture area, determines the thrust/airflow required at the engine face.

THRUST/AIRFLOW ($F/\dot{m}a_0$)



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Analysis based on simple cycle model (which is in accord with detailed cycle deck model GASTURB) shows that ZSE accommodate the thrust requirements of the AFRL aircraft model. The numbers on the last row of the table on vue 11 are in line with the computed curve shown on this vue.

IMPLICATIONS

- Feasibility of zero spill engines
- Large swings in component behavior needed, incompatible with a fixed engine geometry
- Variations needed in
  - Fan pressure ratio, $P_{T1.8}/P_{T2}$
  - Fan inlet axial Mach No, M_2, capability
  - Compressor exit axial Mach No, M_3
  - Compressor pressure ratio, P_{T3}/P_{T2}
  - Turbine nozzle area, A_4
    - Varies by a factor of 4 from M_0=0.8 to M_0=2.5
  - Exit nozzle throat area, A_8

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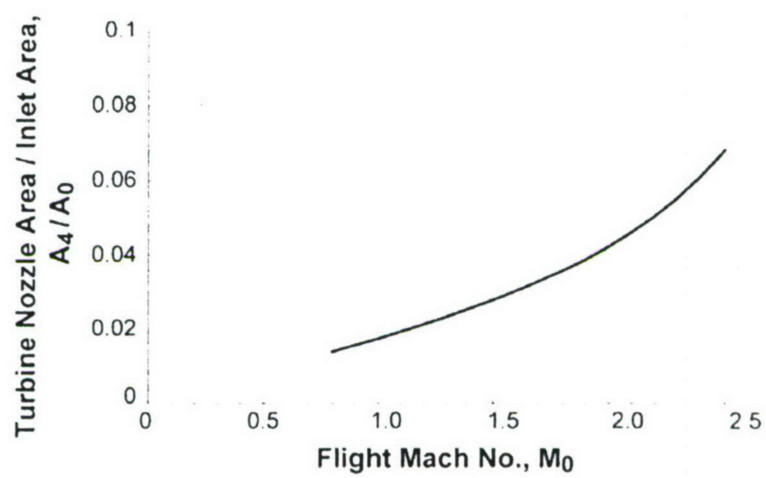
Parametric cycle calculations have been carried out to show how the characteristics of a family of engines would vary with flight Mach numbers subject to the conditions delineated on vue 10. As one would expect the ratios of various area to inlet capture area (A_0) vary widely from subsonic to supersonic flight.

The area ratio in the core, A_4/A_3 (the turbine nozzle area to compressor exit area), varies from about 1.6 to 1.1 as M_0 varies from subsonic to M_0=2.2. This appears to be the range that might be accessible with a variable nozzle of the sort that the Air Force would be interested in. However the variation in A_4/A_2 (turbine nozzle area to compressor inlet area) is much larger, mostly because of the large variation in pressure ratio needed to hold compressor exit temperature constant.

Variation in engine area ratios and specific thrust have been assessed for fan temperature ratio varying from 1.3 to 1.8, compressor exit temperature of 950 and 1089 K, turbine inlet temperature of 1600 and 2255 K. Over the flight number range of 0.8 to 2.5, A_4/A_0 can be expected to vary by a factor of 2 to 5, and the thrust per unit airflow by a factor of 2 to 2.5.

TURBINE NOZZLE AREA A_4/A_0 VARIATION NEEDED



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One potential enabling technology for ZSE to yield a factor of 4 variation in turbine nozzle area is the swirl modulated vaneless variable area high pressure turbine concept as delineated in the next few vues.

INFLUENCE OF SWIRL ON CORRECTED FLOW

- Corrected flow without swirl

$$\frac{\dot{m}\sqrt{T_t R/\gamma}}{Ap_t} = \sqrt{\frac{T}{T_t}}\frac{\rho}{\rho_t}\frac{u_m}{\sqrt{\gamma RT}} = M_m f(M)$$

- Swirl α reduces corrected flow

$$\frac{\dot{m}_{\alpha\neq0°}}{\dot{m}_{\alpha=0°}} = \left[\frac{(\gamma+1)\cos^2\alpha}{\gamma+\cos 2\alpha}\right]^{\frac{\gamma-1}{2(\gamma+1)}}$$

- Thus, can modulate flow by
  - changing area (with swirl fixed)
  - changing swirl (with fixed area)
  - Both

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On this vue: subscript t refers to the stagnation value. M_m is the meridional Mach number and u_m the meridional velocity.

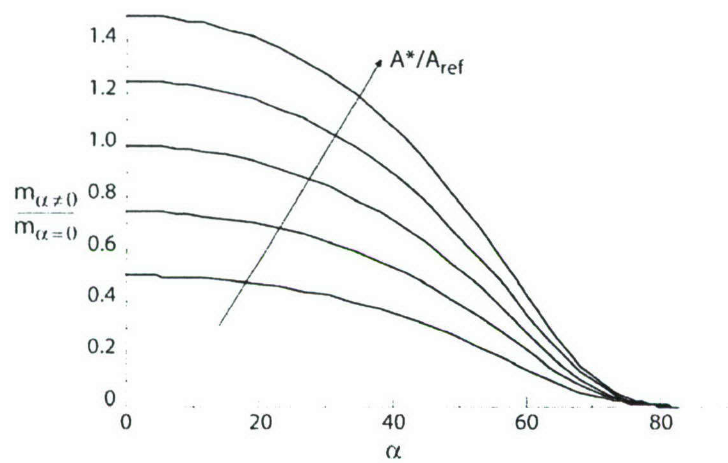
T denotes temperature, ρ the density, p the pressure, R the gas constant, A the area of flow path, γ the ratio of specific heats, M the Mach number and α the swirl angle

$\dot{m}$ the mass flow rate

$\dot{m}_{\alpha=0}$ the mass flow rate corresponding to zero swirl situation

$\dot{m}_{\alpha\neq0}$ the mass flow rate corresponding to a swirl of α

PARAMETRIC VARIATION OF CHOKED MASS FLOW WITH SWIRL ANGLE FOR A* FROM 0.5 TO 1.5 AT 0.25 INTERVAL



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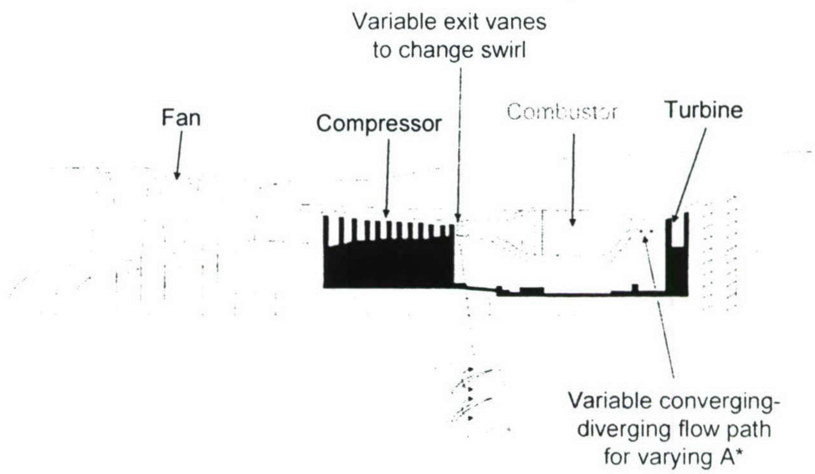
A_{ref} refers to the choked area corresponding to zero swirl.

For given fixed A*, varying swirl angle α from 10 to 60 degree gives a factor of 3.5 change in choked mass flow.

One can achieve a change in choked mass flow by a factor of 11 through a combined variation of swirling angle α from 10 to 60 degree and area A^*/A_{ref} from 0.5 to 1.5.

Thus the above conceptual framework of implementation provides a technological enabler of realizing the zero-spilled engine.

VARAIBLE AREA SWIRLING TURBINE CONCEPT (VAST)



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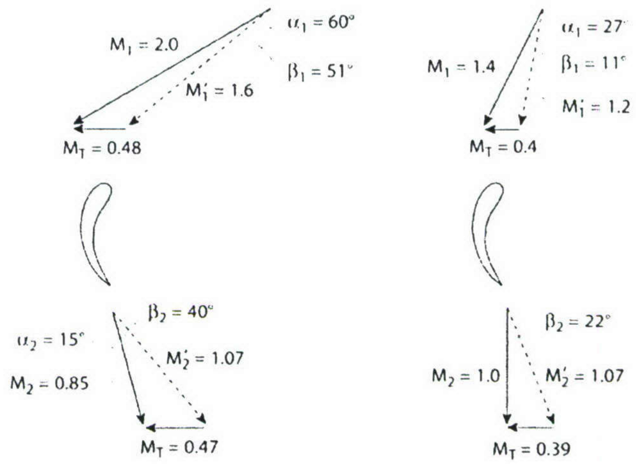
This sketch proposes a conceptual implementation of the modulated swirl vaneless variable area high-pressure turbine (HPT) configuration. The swirl to effect a change in the choked flow is introduced by a variable swirl vane at the compressor exit where the temperature is far lower than that at turbine entry. The effect of the swirl can further be beneficially amplified with a variable (through sliding the inner center body) converging-diverging nozzle between the combustor and turbine. Such a proposed configuration necessitates assessing how the resulting swirl would impact combustor performance and the flow behavior in the converging-diverging nozzle flow path. One concern of using the idea of swirl to alter the choked mass flow is that swirling flow might hinder us from delivering the required flow to the turbine). In a swirling through flow environment within an annular flow path, the endwall flow (i.e. the boundary layer on the hub and casing) can exhibit reversed flow for high swirl angle (a value above 40 degree where such reversed flow might be present in the endwall flow region). Such reversed flow has the potential of altering the flow profile (from the intended one) delivered to the turbine. The accelerating flow in the turbine environment would mitigate this somewhat (i.e. (favorable) axial pressure gradient vs radial pressure gradient). Since the idea entails introducing a swirl upstream of the combustor, we should assess the swirling flow development in a combustor-turbine flow path configuration. Given the limited resource available to the program we have made the decision of analyzing swirling flow behavior in a turbine flow path only. This involves implementing a reasonable set of calculations representative of the intended choking swirling flow environment in the turbine flow path to examine bulk behavior based on the length scales of the problem.

LMT aeronautics will report on the results on the assessment of the effects of swirl on the flow behavior in the converging-diverging nozzle flow path.

EXAMPLES OF DESIGN AT FIXED GEOMETRY
- Preliminary -



PR = 3.8	PR = 1.5
$\dot{m}_{ref} = \mathbf{1.0}$	$\dot{m}^* = \mathbf{3.0}$

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A preliminary mean line design of a turbine to illustrate the change in the turbine characteristics (swing in Mach numbers and flow angles) to be anticipated for a change in swirl angle to achieve a change in the choked mass flow by a factor of 3. While there are attending challenging implications on turbine design, they are not insurmountable. For instance the high inlet Mach number to the turbine can be mitigated through a series of weak oblique shocks (for minimal shock loss) to adjust the flow so that M_m <1 upstream of turbine rotor. To mitigate the high loading requirement, two stage turbine can be used instead of a single stage turbine.

DEPENDENCE OF CHOKED FLOW ON α AND A*
-Summary-

- Choked mass flow dependent on A* and swirl angle α
- Use of swirl angle α and A* to vary choked flow
  - Conceptually feasible, details need further assessment
- For fixed A*
  - Varying α from 10 to 60 gives a factor of 3.5 in choked mass flow
- By varying α from 10 to 60 and A^*/A_{ref} from 0.5 to 1.5
  - Can change the choked mass flow by a factor of 11
- Idea attractive so far
  - Provide substantial leverage for broadening achievable engine variability
- Challenging implications on turbine aero design
  - High inlet flow relative to turbine rotor
  - Interactions with other variable cycle features

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DEFINING AND PLANNING CFD SIMULATIONS OF FLOW IN VAST FLOW PATHS

- Defined a set of CFD calculations to assess response of flow in VAST flow paths to variations in
  - Swirl angle (from 30 to 70 degree)
  - Temperature distribution representative of that at combustor exit that include
    - ➢ Radial variation
    - ➢ Circumferential variation
- LMT Aeronautics will report results from proposed set of CFD simulations

SUMMARY

- A factor of 3 to 6 change in A_4 feasible in VAST
  - A conceptual implementation of VAST was proposed (vue 17)
- VAST a key enabler for
  - Zero spillage engine
  - An adaptable engine optimized for supersonic flight at M = 2 to destination with effective subsonic M = 0.8 loiter
- LMT Aeronautics assessed bulk flow behavior in VAST flow paths excluding the combustor flow path

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VUES FOR FUEL-EFFICIENT ENGINE AND SENSOR-AIRCRAFT

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Within the present effort, the MIT Gas Turbine Laboratory also took the initiatives to examine the requirements of fuel efficient engines for logistic aircrafts and for high altitude sensor aircraft. The key results are summarized in the next few vues.

FUEL EFFICIENCY ENGINES

- **For Polytropic Efficiency=0.92, Thermal Efficiency Maximum at Temperature Ratio=4 to 5**
- **Currently temperature ratio limited to 3 by compressor disk materials**
- **Increasing temperature ratio from 3 to 5 increases fuel specific impulse by about 15%**
- **Design of high-temperature ratio engine architectures**
  - **Very high pressure ratios (up to 300)**
  - **Compressor rim, combustor and turbine Cooling**

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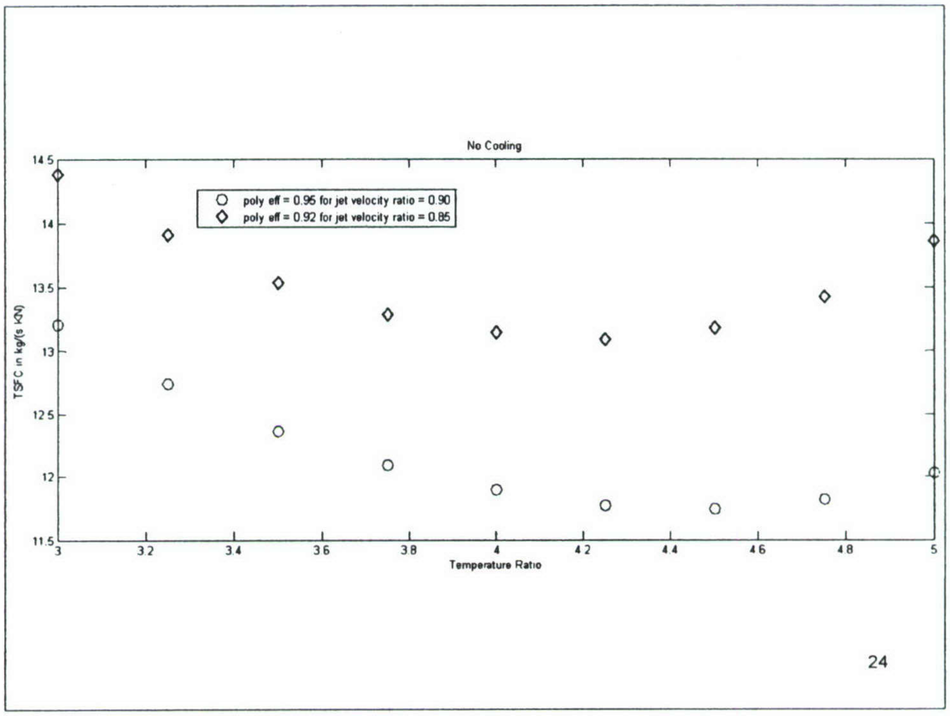
For lowering the fuel consumption of logistic aircraft:
1) High polytropic efficiencies are crucial, e.g for a turbofan with bypass of about 20, the Specific Impulse goes from 8,400 s to 10,000 s for a change in polytropic efficiency from 0.9 to 0.95.
2) When the polytropic efficiency is high,(say 0.95) there is much to be gained from raising the pressure ratio. Calculations indicate that the Impulse peaks at a compressor temperature ratio of about 5, where the bypass is about 20, at a value about 18% above that for a temperature ratio of 3. Of course the compressor discharge temperature for this condition is very high, about 1500 F, necessitating some sort of compressor cooling (cooling of the compressor disc rim)

Overall it seems that progress toward low fuel consumption is likely to come from a means such as the above, which are technically hard but sure to pay off handsomely !



Engine cycle analyses have been implemented to quantify the benefits of improving the polytropic efficiency of compressor from 0.90 to 0.95. The minimum TSFC corresponds to a compressor temperature ratio of 4.5 and the improvement in TSFC is 15%. With compressor cooling additional gain can be expected.

POTENTIAL BENEFITS FROM FEE

- Up to 30% Increase in Range Productivity

- Further Gain if Polytropic Efficiency Can be Improved beyond 0.92

PROPULSION SYSTEM FOR SENSOR AIRCRAFT -REQUIREMENTS AT HIGH ALTITUDE > 60000 ft-

- Combustor pressure ~ 10 atmospheres
  - Compressor pressure ratio over hundred
- Maximum endurance
  - Low SFC
  - Low engine weight
  - Maximum L/D

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Exploratory calculations have been implemented for a sensory aircraft with a design flight Mach number of 0.5 and cruising altitude between 60,000 to 90,000 feet. The combustor pressure is taken to be ~10 atmospheres and the compressor pressure ratio is over hundred. The compression system consists of Fan-LPC-HPC system with intercooling between LPC and HPC; the HPC exit temperature is set at 900 K.

The criteria used are minimum SFC (corresponding to $V_{bypass}/V_{nozzle} \sim \eta_{fan} \times \eta_{LPT}$), low engine weight and maximum L/D.

PRELIMINARY ENGINE WEIGHT ESTIMATION
- Excluding Intercooler Weight -

Altitude	Engine mass/Fan Area (kg/m^2)	Thrust/(engine weight)	Endurance
60,000 ft	875	0.74	26 hr
70,000 ft	1047	0.47	16 hr
80,000 ft	1526	0.2	11 hr
90,000 ft	4000	0.04	?

$$Endurance \sim \left(\frac{L}{D}\right)\frac{1}{g \times SFC}\, ln\left(\frac{W_i}{W_f}\right)$$

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The results shown in the table indicate the variation in engine mass per unit fan area, thrust to engine weight ratio and the endurance one might expect for various cruising altitude. The present exploratory calculations show that the ceiling for sensory craft is about 80,000 ft, a limitations due to reynolds number effect.

LESSONS LEARNED FROM CYCLES STUDY OF PROPULSION REQUIREMENTS FOR SENSORY CRAFTS

- Ceiling for sensory aircraft ~ 80,000 ft
  - Limitation by Reynolds number effect
    - ➢ Potential solution: aspiration
- Technology enablers
  - A new invention for intercooling
  - Aspiration

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OVERALL SUMMARY

- Significant gain in performance realizable with appropriate adaptability/variability engineered into aircraft turbine engines
- Feasibility of zero spillage engine with substantial attendant benefits
- Fuel efficiency engines achievable with very high-pressure ratio (~100s) compression system
- Enabling technologies needing future research/development effort
  - VAST for substantial variability (in ZSE)
  - Compressor rim cooling and intercooling
  - Aspiration

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The analyses and calculations that have been carried out show that significant gain in performance (as measured in terms of thrust specific fuel consumption and range productivity) can be obtained by having variability and adaptability in aircraft turbine engines compared to those with fixed geometry. A 5-10% variability in (turbine and propelling) nozzle areas would allow the compression system to have an operating point for subsonic loiter identical to that for supersonic flight to destination, resulting in a 20% improvement in thrust specific fuel consumption. The ability to engineer variability/adaptability into engines would be an enabler for realizing zero spillage engine with significant attendant benefits (see vue 9 note page). Progress toward low fuel consumption (i.e. fuel efficient engine technology) can be achieved through engineering very high pressure ratio (~ hundreds) compression system with high polytropic efficiency compressor components.

There is thus a strong incentive to research and develop enablers for the realization of engines with appropriate variability and adaptability, zero spillage engine and fuel-efficient propulsion systems. These enablers include VAST that we put forward, technologies for compressor rim cooling, intercooling and flow aspiration. While the realization of these technology enablers are technically challenging in practice but they are sure to pay off handsomely (such as significantly broadening the scope and flexibility of missions presently not accessible with engines of fixed geometry) !